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A NOTE ON THE GENERAL SCALING OF HELICOPTER BLADE-VORTEX INTERACTION NOISE

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F.H. SCHMITZ , D.A. BOXWELL , S. LEWY , C DAHAN

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A NOTE ON THE GENERAL SCALING OF HELICOPTER BLADE-VORTEX INTERACTION NOISE

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Abstract

A model-rotor acoustic experiment in a three meter open section anechoic wind tunnel (CEPRA-19) is described. The important scaling parameters are reviewed and some on-line acoustic data are presented for conditions known to produce blade-vortex interaction noise on a single rotor helicopter. Time averages of the model-scale acoustic pulses are compared with similar full-scale data taken in-flight under the same non-dimensionalized conditions. Good general agreement between model and full-scale pulse shapes and amplitudes of blade-vortex interaction noise is shown for one advance ratio over a range of inflow conditions for the rotor. Directivity sweeps for a condition known to generate blade-vortex interactions are presented. Some model rotor testing limitations of the CEPRA-19 facility in its present configuration are indicated.

Introduction

The design and operational problems associated with helicopter external noise have recently been highlighted with the promise/threat of external noise regulations. While the debate continues as to the pros and cons of noise regulations, the helicopter community has been assessing its ability and confidence to design to definitive noise limits. A major finding of this assessment is that theoretically predicting helicopter external noise to within two to three PndB is not yet possible [1]. More research into the fundamental mechanisms of rotor noise and its relationship to the full-scale helicopter acoustic signature is needed. Due to the long term nature of this basic approach, the assessment also suggested the use of model- and full-scale acoustic testing. Unfortunately, full-scale envelope testing of rotors in an anechoically acceptable environment is quite expensive, difficult, and time consuming. The more cost effective approach is to use scale model testing in acoustically treated wind tunnels. However, it is first necessary to demonstrate that the model- scale events (aerodynamic and acoustic) do, in fact, represent the full-scale acoustic events of interest.

In this paper, the verification of model- to full-scale acoustics is facilitated by considering one of the more intense and distinctive helicopter sounds , blade-vortex interaction impulsive noise. As shown in the literature[2-5], when impulsive noise exists its "popping" or "cracking" sound tends to dominate the radiated noise of a helicopter. This noise is known to originate from two distinct aerodynamic events : high tip Mach numbers on the rotor's advancing blade that cause near-transonic disturbances to radiate high-speed impulsive noise, and blade-vortex interaction on the advancing and retreating sides of the rotor disc that radiates impulsive sounding noise.

Through the research of the past few years, much is known about high-speed impulsive noise radiation. In-flight acoustic data[2] have shown that high-speed impulsive noise radiates predominantly in the plane of the rotor disc and in the direction of the helicopter's flight path. Model-to full-scale acoustic comparisons have demonstrated that high-speed impulsive noise can be quantitatively studied on aerodynamically scaled models. The two most important parameters are geometric rotor scaling and advancing-tip Mach number. When these are duplicated, the measured time histories of high-speed impulsive noise on model rotors duplicate the full-scale event[6]. It has also been theoretically shown that below the "delocalization Mach number", linear theory adequately predicts the general shape but underpredicts the amplitude of the acoustic radiation[7-9]. However, above the "delocalization Mach number", the non-linear aerodynamic near field is needed to adequately predict the character and intensity of the noise[10-12].

Much less is known about the aerodynamics and acoustics of blade-vortex interaction impulsive noise. A necessary condition for acoustic radiation is the close passage of a blade with a previously trailed tip vortex. The resulting interaction causes a fluctuating pressure on the rotor blade that radiates noise. Under some conditions, theoretical models[13,14] have shown promising agreement with data. However, under typical descent conditions for a full-scale single rotor two-bladed

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helicopter, theory and experiment have shown significant discrepancies even when measured blade pressures were utilized in a non-compact theoretical formulation[15]. Full-scale testing has indicated that the problem is aerodynamically complex because of the variability of the wake system and the possibility that unsteady transonic aerodynamics influences the interaction[2,3,16,17]. There have also been many model-scale tests performed to help understand the blade-vortex interaction problems [18-20]. Parametric studies of rotor flow conditions and of the effectiveness of different rotor tips have been explored[21].

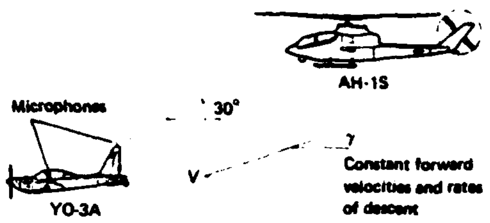

Fig. 1 – Full-scale blade-vortex interaction acoustic measurement technique.

The major objective of this paper is to demonstrate that blade-vortex interaction noise is a scalable phenomenon. A description of the model scale experiment, the non-dimensional parameters which are necessary to duplicate full-scale acoustics on a model rotor, and the directivity of the radiated noise will be presented. Some of the successes and problems associated with the acoustic testing of model rotors in the CEPRA-19 anechoic wind tunnel will also be reported.

Full Scale Data

The full-scale data used in this comparison between the model- and full-scale was gathered using the in-flight technique developed by the Aeromechanics Laboratory[22]. A quiet fixed-wing aircraft (YO-3A) was flown in formation with the subject helicopter (AH-1) (Fig. 1) and stationary acoustic data was gathered over a full range of flight conditions. For blade-vortex interaction noise, selected conditions of forward velocity and rate of descent were flown with the microphone positioned directly ahead of the helicopter but thirty (30) degrees below the rotor's tip-path-plane. The major advantages of this technique are the long data records, absence of ground reflections, and the ability to fly conditions normally associated with helicopter terminal area operations. Although the procedure has been used to evaluate the noise radiation of many helicopters, the full-scale data presented in this paper was taken on an AH-1S helicopter in 1978 and 1979 on two separate test programs and reported in Refs. 16 and 23. The data is repeatable and clearly defines the blade-vortex interaction noise phenomena of interest.

CEPRA-19 Anechoic Wind Tunnel

Acoustic testing of the rotor model was conducted in an anechoic wind tunnel located in France at CEPr (Center for Propulsive Studies) near Paris. This facility, CEPRA-19, shown in Fig. 2[24] is of the open circuit, open test section design and is one of the largest anechoic wind tunnels in the world. A concrete test chamber surrounds the open jet and is approximately one quadrant of a sphere with a 9 meter radius. The walls and floor are lined with acoustic wedges 1 meter in length, giving an acoustic cut-off frequency of approximately 200 Hz. Both the entrance and diffuser portions of the tunnel circuit are anechoically treated with acoustic baffles to protect the test chamber from exterior and tunnel drive system background noises.

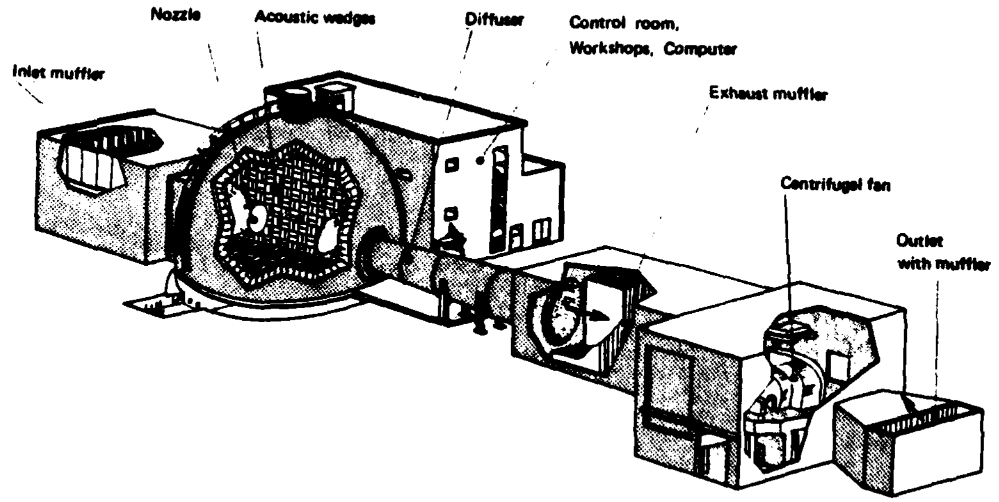

Fig. 2 – A schematic of CEPRA-19 anechoic wind tunnel.

The free jet is 12 meters long. A circular 3 meter diameter nozzle was installed for these tests which theoretically allows a maximum jet velocity of 60 meters/second. The resulting large mass flow rate is collected by a large solid fiberglass collector and extracted by a centrifugal pump to the outside through the acoustic baffling system. The specific design used for these tests was developed using an aerodynamic small scale model[25]. In model-scale it was shown that the 3 meter configuration would perform satisfactorily. It was also reported that potential core mean velocity profiles showed no anomalies and turbulence intensity was lower than 1%. Both the nozzle and collector were of hard-surface fiberglass construction and the nozzle lip was treated with 10 cm of serrated acoustic foam. An adjacent control room housed all measurement instrumentation and wind tunnel/rotor drive controls.

The background noise of the CEPRA-19 wind tunnel posed no problem for the present tests. In general the noise of microphones inside the flow (with nose cones) or exterior to the flow (with wind screens) was quite low in intensity above 30 Hz.

Rotor Test Stand

The rotor model was mounted on the Aeromechanics Laboratory's rotor test stand that was installed in the CEPRA-19 test section as shown in Fig. 3. The stand housed a 400 cycle variable frequency electric motor that provided 90 HP to drive the test rotor up to speeds of 3000 RPM. For all test conditions, the ambient drive system noise was below the wind noise floor and thus not a significant contributor to the measured model rotor noise. Rotor shaft signals, at rates of 1, 60, and 180 per revolution, were generated by the stand. The test rotor was mounted on a teetering hub assembly with collective and longitudinal and lateral cyclic rotor controls provided by remotely controlled electric swashplate actuators. The rotor hub and swashplate assembly were mounted on a six-component strain gauge balance located in the top section of the rotor stand. Drive shaft torque was also measured. As shown in Fig. 3, the upper portion of the rotor test stand that extended into the jet flow was shrouded with an aerodynamic fairing and wrapped with 2.54cm of acoustic foam. The rotor stand was mounted to the concrete floor of the test section resulting in a rotor hub center 0.445 meter above the tunnel jet centerline and 2.677 meters from the nozzle lip. The fore and aft position of the rotor is illustrated in Fig. 4 and was chosen to 1) enhance the collection of combined jet and rotor wake, 2) place the rotor inside the theoretical jet shear layer boundaries, and 3) allow placement of microphones at the desired scaled distances from the rotor hub. The rotor direction of rotation was chosen so as to place the advancing blade on the more spacious side of the test chamber.

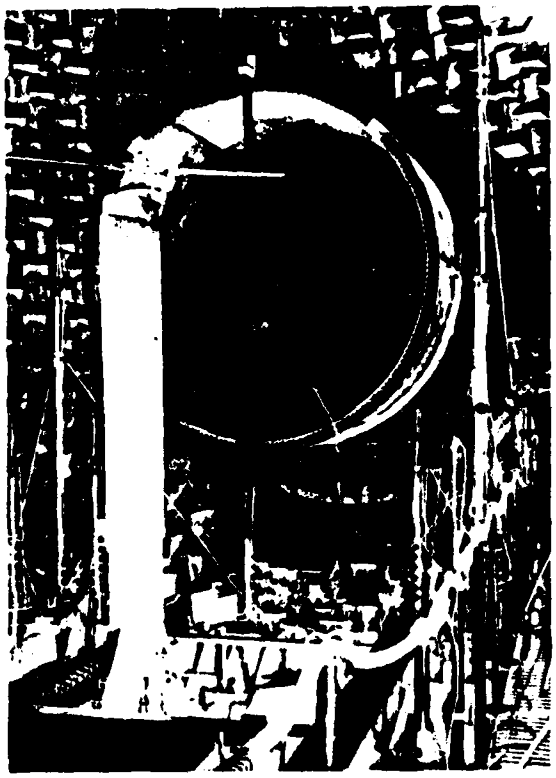

Fig. 3 — 1/7 scale OLS model rotor and test stand installed in french CEPRA-19 anechoic wind tunnel.

Acoustic Instrumentation

Fifteen 1.27 cm (0.5 inch) B & K 4133 microphones were spacially located around the rotor. Initial microphone locations are as illustrated in Fig. 4, typically 3.26 meters from the rotor hub. This distance corresponds to scaled microphone positions during the full-scale in-flight U.S. Army acoustics investigations referenced earlier[16,23]. During the course of the wind tunnel tests, some of the microphones were moved in accordance with online analysis and for directivity investigations.

Referring to Fig. 4, microphones 1 - 4 were located in the jet flow, mounted to the nozzle centerline strut, pointed into the flow and equipped with nose cones. Microphone 1 was in-plane with the rotor hub and microphone 4 was located 30° below that plane. Microphones 5 - 7 and 9 - 15 were located outside the jet shear layer, held by floor mounted struts, fitted with standard protective grid plus foam wind screen, and pointed towards the rotor hub. Microphone 8 was similarly outfitted and oriented, but was mounted to a movable facility survey apparatus and located above the rotor, on axis outside the jet.

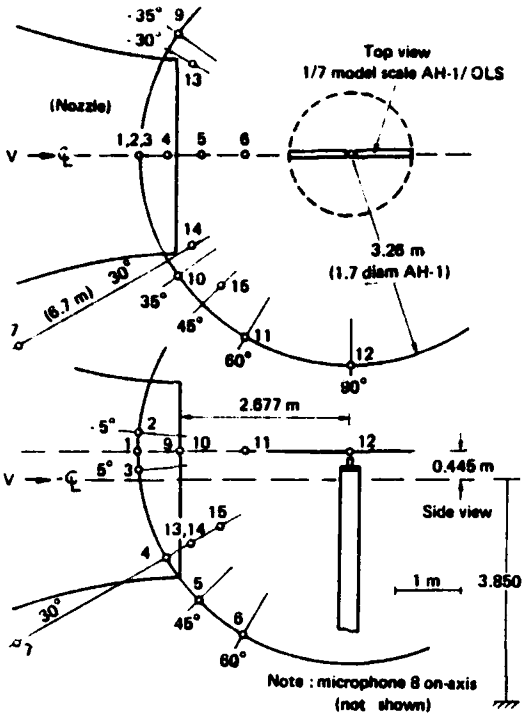

Fig. 4 – Location of rotor and microphones (initial configuration) in CEPRA-19 open jet wind tunnel.

Signals from each microphone were monitored on-line for proper gain/attenuation and simultaneously recorded on both a 32 and 28 channel FM tape recorder set for 76.2 cm/sec and 20 kHz frequency response. Rotor azimuth signals, flapping potentiometer output, balance data, and time code also were all recorded along with the acoustic data. During the one minute tape recording, an HP 5451B time-series analyzer was used to generate representative and averaged (100) time histories and power spectra of 4 channels of acoustic data. It is, for the most part, this on-line digitized data that are used in the results of this paper.

Wind tunnel velocity, temperature, and dewpoint as well as rotor speed, swashplate control inputs, and balance information were processed on-line using an Aeromechanics Laboratory HP 85 computer. Rotor performance in terms of advance ratio, thrust coefficient, hover/advancing-tip Mach number, and tip-path-plane angle was computed prior to and during each run.

Scaled Model Rotors

During the model experimental test program, a number of different rotors were investigated. The primary rotors were 2-bladed, 1/7 geometrically scaled models of the AH-1G/OLS main rotor, with and without pressure instrumented blades. Secondary rotors were 1) the AH-1G/OLS/747 and AH-1G/OLS/FX098 modifications of the tip shape and airfoil section, respectively, 2) a UH-1H model rotor, used for baseline checks, and 3) a 4-bladed rotor system provided by SNIAS. Only on-line acoustic results from the uninstrumented model AH-1G/OLS rotor are presented in this paper.

Geometric characteristics of the model AH-1G/OLS rotor are shown in Fig. 5.

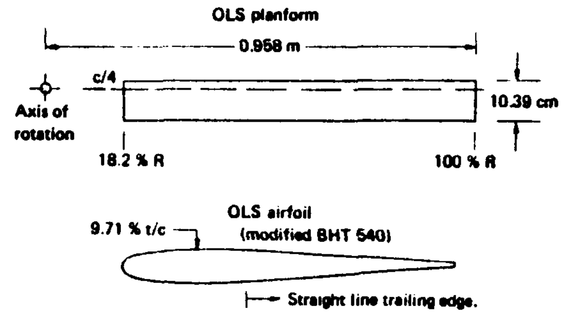

Fig. 5 – Geometric characteristics of 1/7 scale model AH-1G/OLS rotor.

The rotor is 1.916 meters in diameter, has a 10.39 cm chord, and is twisted -10 degrees (washout). The AH-1G/OLS is a rectangular tip BHT 540 rotor blade which has been increased in thickness and chord slightly to accomodate full-scale pressure instrumentation[17]. The resulting special sections OLS airfoil has a 9.71% thickness instead of the 9.33% thickness of the standard 540 rotor. Model rotor blade construction is an aluminum alloy spar with balso/hardwood section wrapped by fiberglass skins.

Test Conditions

Besides blade geometry and relative orientation between the rotor and the microphone, there are at least four non-dimensional parameters which should be duplicated if model-scale acoustic data are expected to match full-scale data. First, advance ratio, defined to be the ratio of the helicopter's forward velocity of translation divided by the main rotor tip-speed (i.e. $\mu = \frac{V}{V_{TIP}}$), governs the large-scale geometry of the blade-vortex interaction phenomena. When viewed from above, the rotor appears to slice through the epicycloidal pattern of shed tip vortices. The resulting loci of interactions play an important role in determining the number and strength of the blade-vortex interaction encounters and thus strongly influences the radiated noise[15]. Second, as in most of rotor acoustics, rotor tip Mach number also has a large, if not the largest, effect on noise radiation and thus must be matched in a scaling experiment. Because μ is also fixed, the advancing-tip Mach number ($M_{AT} = (1 + \mu)M_{HT}$) of model and full-scale is automatically scaled. In effect, all velocities associated with the large-scale geometry of the blade-vortex interaction encounter are governed by μ and M_{HT}.

The thrust coefficient divided by rotor solidity, C_T/σ, is the third parameter of importance. It governs the local angle of attack of the rotor and thus generates similar local blade pressures on the model- and full-scale rotors. For a geometrically scaled test, the rotor solidity is the same on model and full-scale, making it necessary to match the thrust coefficient, C_T, of both rotors. The fourth necessary non-dimensional parameter is the non-dimensional inflow velocity distribution ratio ($\lambda = \mu(-\alpha_i + \alpha_{TPP})$) at the position of the vortex interaction encounter. It governs the local vertical separation between the vortex and the blade at the time of an encounter. In a rigorous sense, this parameter should scale over the portion of the rotor disc where blade-vortex interactions occur. However, it is often assumed that by scaling geometric properties and C_T, an average value in space and time of the induced angle (α_i) at the rotor disc governs the interaction problem ($\alpha_i \sim \frac{C_T}{\mu}$). Therefore if C_T and μ are duplicated in a model- to full-scale test, the tip-path-plane angle (α_{TPP}) becomes the fourth non-dimensional test variable.

In normal unaccelerated level flight, the full-scale helicopter pilot must tilt the rotor tip-path-plane (α_{TPP} = the angle between the plane of the rotor tips and the incoming velocity vector ; positive for rearward tilt) to balance the drag of the vehicle at each velocity. The result is an increasingly negative tip-path-plane angle as shown in Fig. 6 for the AH-1 helicopter considered in this test. In a climb, the rotor must be tilted further forward ($-\alpha_{TPP}$) both to balance drag and to oppose gravity, whereas in a descent the rotor must be tilted rearward ($+\alpha_{TPP}$). The strongest blade-vortex interactions are known to occur in the descent condition when the tip-path-plane angle is positive forcing sections of the shed tip vortices close to or into the rotor's tip-path-plane.

The cross-hatched lines shown in Fig. 6 describe the general boundaries of the model experiment. Most of the test points were confined to low advance ratios because of the limited speed capabilities of the CEPRA-19 tunnel. Advance ratios below 0.1 were not chosen because of the uncertainty of the flow in the open jet tunnel. The dark solid line in Fig. 6 spans the sequence of conditions which was selected for comparison with flight data in this paper. The sequence is representative of a 60 knot (IAS) full-scale condition (μ =.164) and consists of a data map from level flight to 900 ft/min rate of descent. For each condition in the sequence, α_{TPP} of the model rotor was varied to move along this line and thus match the desired full-scale conditions.

The relatively large size of the rotor (1.91 meters), when compared with the size of the open jet test section of CEPRA-19 (3.0 meters), alters this map to some degree. Free-jet wall corrections tend to decrease the tip-path-plane angle (α_{TPP}) of the model-scale blade-vortex interaction noise , the largest correction occuring at low μ and high C_T. Unfortunately, the wall correction calculations for a rotor in this environment are not trivial. The vortex structure of the rotor wake should be bounded by a complex image system of vortices so arranged that at each instant of time the static pressures at the jet boundaries are similar to that of the unbounded problem. A complete treatment of this unsteady problem has not yet been attempted for CEPRA-19. Instead, a very simplified fixed-wing wall correction estimation was calculated based on Milne-Tompson[26]. The closed form result indicates that the space and time averaged wall correction ($\Delta\alpha_{TPP}$) for this rotor with a C_T = .0056 and μ = .164 is about -1.5° to -2°. Therefore to match model-and full-scale data, the α_{TPP} model scale values should be decreased by 1.5° to 2.0°.

As explained previously, the model rotor was chosen to be a replica of the AH1/OLS blade tested in Refs. 3 and 17, not the AH-1/540 rotor. Scaled geometric differences between the two airfoils are small and are not thought to be important for the data presented here. However, great care was taken to insure accurate readings of hover tip Mach number, M_{HT}. Rotor RPM was measured by three independent measurement devices and checked periodically during the test. Local speed of sound changes were calculated accounting for changes due to water vapor. An estimated accuracy of ± .001 for M_{HT} was achieved for most of the test conditions. Care was also taken with rotor thrust measurements. The on-line balance data was compensated for temperature yielding an estimated accuracy of ± 2%.

One additional point should be made with regard to scaling. No attempt in this experiment was made to dynamically scale the rotor blades. Instead, the experiment was designed for a relatively stiff rotor with the hope that aeroelastic effects would be small. Fortunately, the results to date indicate that this is a good assumption. However, this is not thought to be generally true for other configurations which are softer in torsion and flap than the AH-1 rotor.

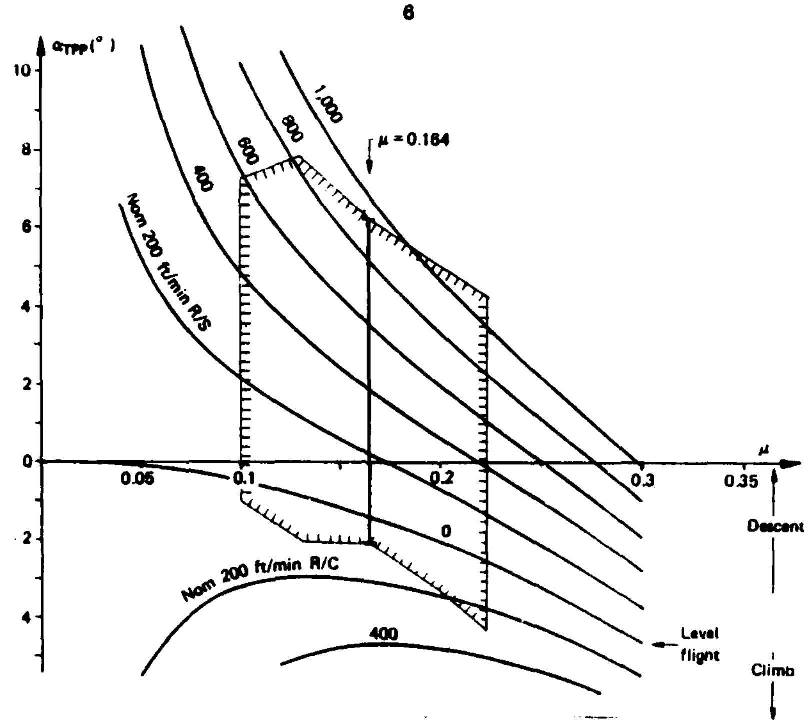

Fig. 6 – Comparison of full-scale and model-scale test conditions.

Time History Comparisons

Perhaps the most rigorous test of the scalability of impulsive noise is the most direct –that of simply comparing the character of the model- and full-scale acoustic time histories on a one-to-one basis. In addition to being a straightforward comparison, it is also helpful in identifying the occurances of blade-vortex interactions in the acoustic signatures. This phenomalogical approach is illustrated in Fig. 7 for the AH-1G helicopter[23] for a microphone located approximately 30 degrees beneath the plane of the rotor tips. This relative orientation of the microphone and the rotor is known to maximize the blade-vortex interaction noise[2] and to reduce the intensity of high-speed impulsive noise. In part (a) of this figure, a measured acoustic time history is shown for one rotor revolution as measured using the full-scale in-flight technique. The helicopter and measurement aircraft were flown in formation at a 60 knot (IAS) partial-power descent (400 ft/min rate of descent), a condition known to produce blade-vortex interaction noise. The four important non-dimensional scaling parameters discussed in the previous section of this paper are also listed in part (a) of this figure. At this 30 degree microphone position, both the blade-vortex interaction noise and high-speed impulsive noise are discernable. During advancing blade-vortex interaction, a sequence of narrow, small-negative and large-positive spikes occurs in the waveform just before the more broad negative pressure pulse. The latter is the high-speed thickness noise and is characteristic of acoustic radiation below the "delocalization Mach number" at a microphone position 30° under the rotor plane. It should be noted that the full-scale data shown in Fig. 7 is representative snapshot of a long data record. Time averaging of the in-flight signals is prohibited by small but discernable changes in the relative position between the microphone and the helicopter. However, an attempt was made to optimize the signal-to-noise ratio by choosing a representative snapshot which did not have a tail-rotor impulsive noise spike contaminating the phenomena of interest.

The model-scale data, at approximately the same scaled geometric distance, is shown in parts (b) and (c) of Fig. 7. Part (b) presents one instantaneous snapshot of one rotor revolution while part (c) presents an average of 100 revolutions. The similarity of the model- and full-scale pulse shapes is evident. The relatively wide negative "thickness" pulse is preceded by two sharp blade-vortex interaction pulses on the model-scale data. Average peak amplitudes as well as the character of the events are duplicated quite well on the model-scale rotor. However, in a more critical light, the pulse width of the high-speed impulsive noise is not duplicated very well at this 30 degree microphone position. In fact, the pulse width is about 50% wider than full-scale. As discussed later, this is thought to be caused by either reverberation/reflection effects at the microphone 4 location near the hard wall of the nozzle and/or free jet shear layer effects. Reverberation/reflection ef-

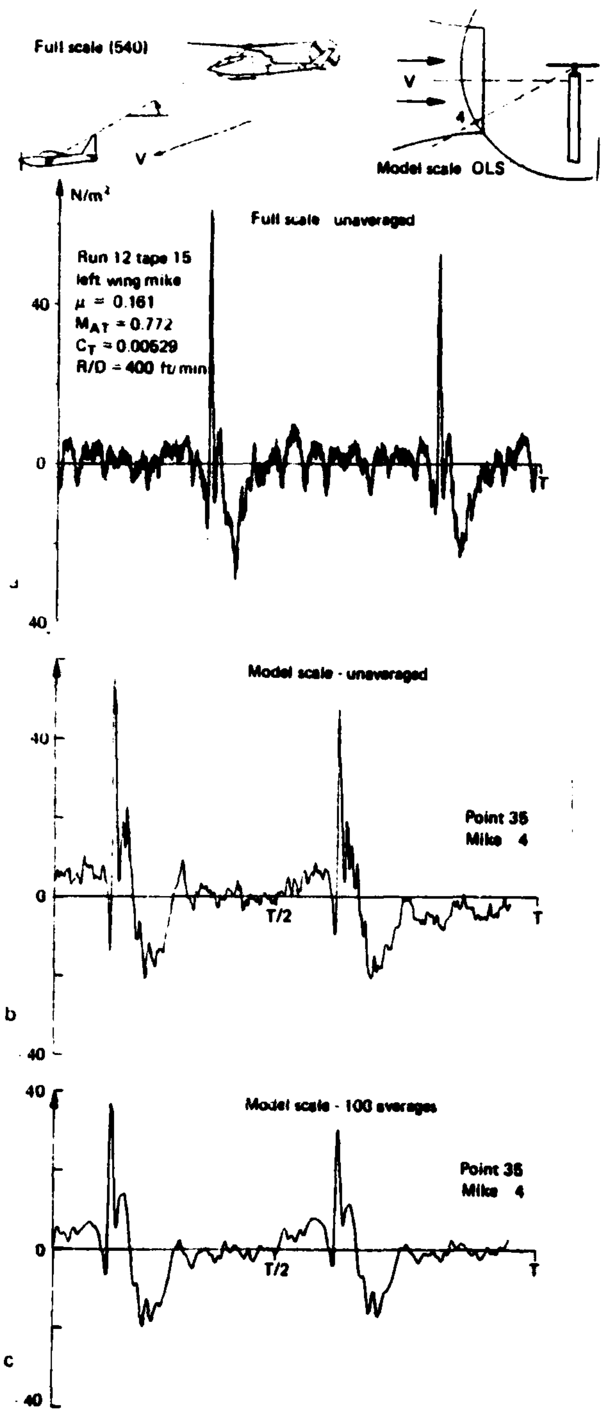

Fig. 7 – Time history comparisons of full-scale and model-scale helicopter blade-vortex interaction noise.

fects are also believed to be responsible for some of the averaged periodic noise on the model-scale data.

A more complete picture of the comparison between model- and full-scale blade-vortex interaction noise is shown in Fig. 8 as a function of the rate of descent (for full-scale data) and tip-path-plane angle (for model-scale data). As discussed previously, approximately a 1.5° wall correction should be subtracted from the model tip-path-plane angle to match full-scale inflow conditions. Good general agreement between model-scale and full-scale is apparent at all descent conditions. The averaged model-scale blade-vortex interaction amplitudes as well as pulse shapes match the full-scale data. The figure also shows that the acoustic pulses which are oldest in time tend to decay while those pulses which are youngest in time tend to grow with increasing descent rate (more positive tip-path-plane tilt). As explained before, as the tip-path-plane of the rotor is tilted rearward, simulating descending flight, a vortex-interaction encounter with an older vortex (youngest in time) is more likely. The model- and full-scale data substantiate this trend and demonstrate the continuity of the blade-vortex interaction process. As noted in this figure, the 30° microphone position (directivity angle) also changes slightly during the increasing tip-path-plane angle sweep. This is an unavoidable result of microphone positions which are fixed in space in the model-scale testing.

Figs. 7 and 8 also illustrate some of the problems of testing rotors in the CEPRA-19 facility. The high-level periodic noise shown on the model-scale data is due partly to the reverberation/reflection of the three-meter nozzle. The rotor impulsive noise is believed to reflect off of the nozzle and to drive the nozzle structural frequencies thus providing the lower level, yet contaminating, background noise. Because this was the first test of a rotor in this facility, no attempt was made to alter the nozzle design to eliminate the problem. A 2-to 3% turbulence level was also measured in the potential core of the jet - just before it passed the rotor blade positions. As a result the rotor blades had a tendency to wobble about 1/3 of a degree in tip-path-plane angle which was visible on the stroboscopic tracking device. This rotor blade unsteadiness is indicative of inflow unsteadiness which makes the blade-vortex interaction process unsteady as well. Consequently, the time-averaged pulses shown in Fig. 8 do not have the sharp character of the full-scale data nor do they have as large an amplitude. Lower inflow turbulence levels would undoubtedly improve this situation and should be considered in future testing of model-scale rotor acoustic tests.

Model-Scale Directivity

One of the most interesting aspects of the blade-vortex interaction phenomena is the directivity of the resulting noise. In general, it had been reported that the noise is radiated forward and down approximately 30° beneath the rotor

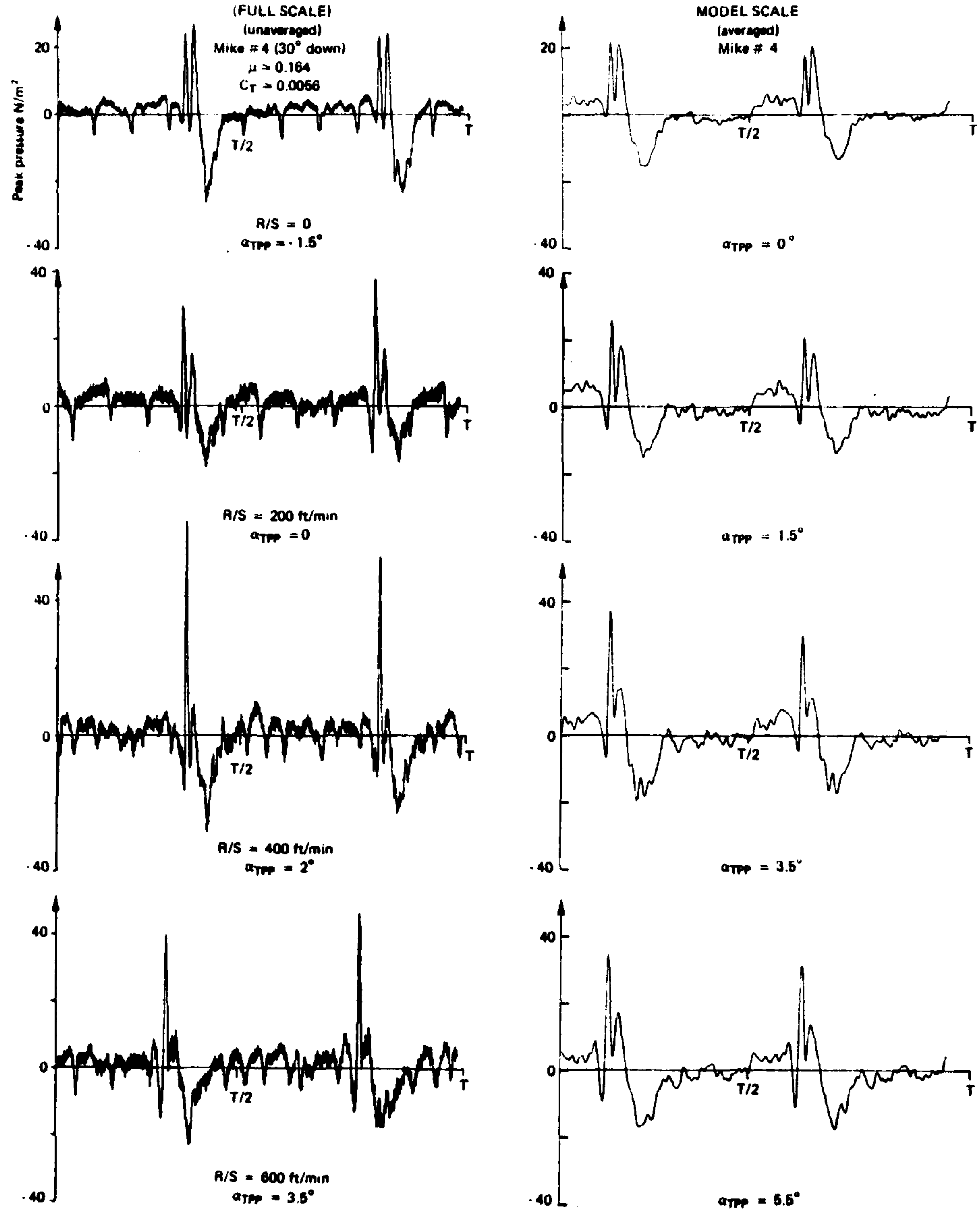

Fig. 8 – Time history comparisons of full-scale and model-scale helicopter blade-vortex interaction noise for several descent conditions.

plane[2,3,18]. Fig. 9 confirms this fact for the μ = .164 and α_{TPP} = 3.5° case but also indicates that the rotor noise decay is less gradual from the 30° position toward the rotor's tip-path-plane than further below this position.

The lateral directivity sweep shown in Fig. 10 at the ψ = 30° elevation angle reveals many interesting phenomena as well. The impulsive noise spike is quite weak at the θ = 90° position of the microphone and appears to consist of one event. It grows as the microphone is moved to the 45° position and grows even larger at the θ = 30° position. Here it is apparently joined by another pulse. The amplitude of the first pulse increases further at the straight ahead microphone position and remains high at the microphone retreating side positions. At the -45° position, a decrease in amplitude occurs, some of which may be attributable to blockage by the rotor stand.

During the test, many other microphone locations were tried in an attempt to locate other directivity lobes of the advancing blade-vortex interaction problem. As the testing progressed, the poorer (less amplitude) locations were abandoned and the microphones were moved to a noisier location. At the end of the testing, most of the microphones were generally located ahead of the helicopter, between an elevation angle of 0-to 45° under the rotor's tip-path-plane. This observation is in agreement with the data shown in Figs. 9 and 10.

Some caution should be exercized when interpreting the data recorded on the microphones located near the 3 meter nozzle. The reverberant/reflected field could have distorted the results somewhat. Shear layer effects can also be important at these microphone locations because the sound path enters the shear layer at shallow angles. Nevertheless, the general trends illustrated in Figs. 9 and 10 are believed to be correct.

Some Testing Limitations

Due to difficulties in data interpretation only a small portion of the data gathered in this joint acoustic test has been reported in this paper. Additional work is planned showing the sensitivity of the model data to parametric and rotor blade changes. Nevertheless, the experience gained to data in the process of preparing this paper can help highlight some important acoustic testing considerations.

One consideration is the adequacy of the aerodynamic environment in open jet anechoic tunnel testing. As measured in model scale[25], the turbulence intensity of the CEPRA-19 jet core was anticipated to be about 1%. RMS intensity levels of the full-scale wind tunnel have indicated levels of 2-to 3% without the rotor present. New results on a two meter nozzle in CEPRA-19 have also indicated a predominance of flow variation in the vertical plane. These pieces of information explain some of the phenomena which occured during the testing. The resulting large rotor angle of attack variations are believed responsible for the wandering of the tip-path-plane during the testing. The same angle of attack variations are also believed to be responsible for much of the model-scale unsteadiness in the acoustic data. It remains to carefully document the unsteadiness of the CEPRA-19 tunnel in the 3 meter configuration so that the remainder of the acoustic and blade pressure data can be correctly interpreted.

A second important consideration is the extent and magnitude of reflection/reverberation problems. The CEPRA-19 tunnel is surrounded by a chamber which is anechoic above 200 Hz. However, for helicopter acoustic testing, many of the microphones were, by necessity, located near the tunnel nozzle. The nozzle apparently reverberated when rotor impulsive noise impinged on it. The actual pulse that was measured on the microphones in or near the nozzle was the sum of reflected and reverberated waves in addition to the signal of interest. Although possible reflections were diagnosed early in the test by impulse-charge testing, the role of combined reflection and reverberation distortion of the data was underestimated. It is imperative in future testing that all major surfaces near microphones of interest be treated to avoid these problems.

The size of the model-scale rotor is another factor that deserves some comment. Aerodynamically scaled rotors tend to be 1/5 to 1/10 of full-scale for many helicopters. This implies that the frequencies of the model are 5 to 10 times the full-scale. Therefore, if a 2,000 Hz full-scale component is to be scaled, it will occur in model scale at 10,000 to 20,000 Hz. Unfortunately microphones become more directional at about 10,000 to 20,000 Hz and thus quantitative interpretation becomes increasingly difficult. Size also works against the successful testing of rotors in an open jet wind tunnel. If the rotor diameter is large compared with the nozzle diameter, large inflow corrections need to be made for the thrusting rotor. In addition, large rotors operate in the outer edges of the potential core which tend to have more unstable flow characteristics.

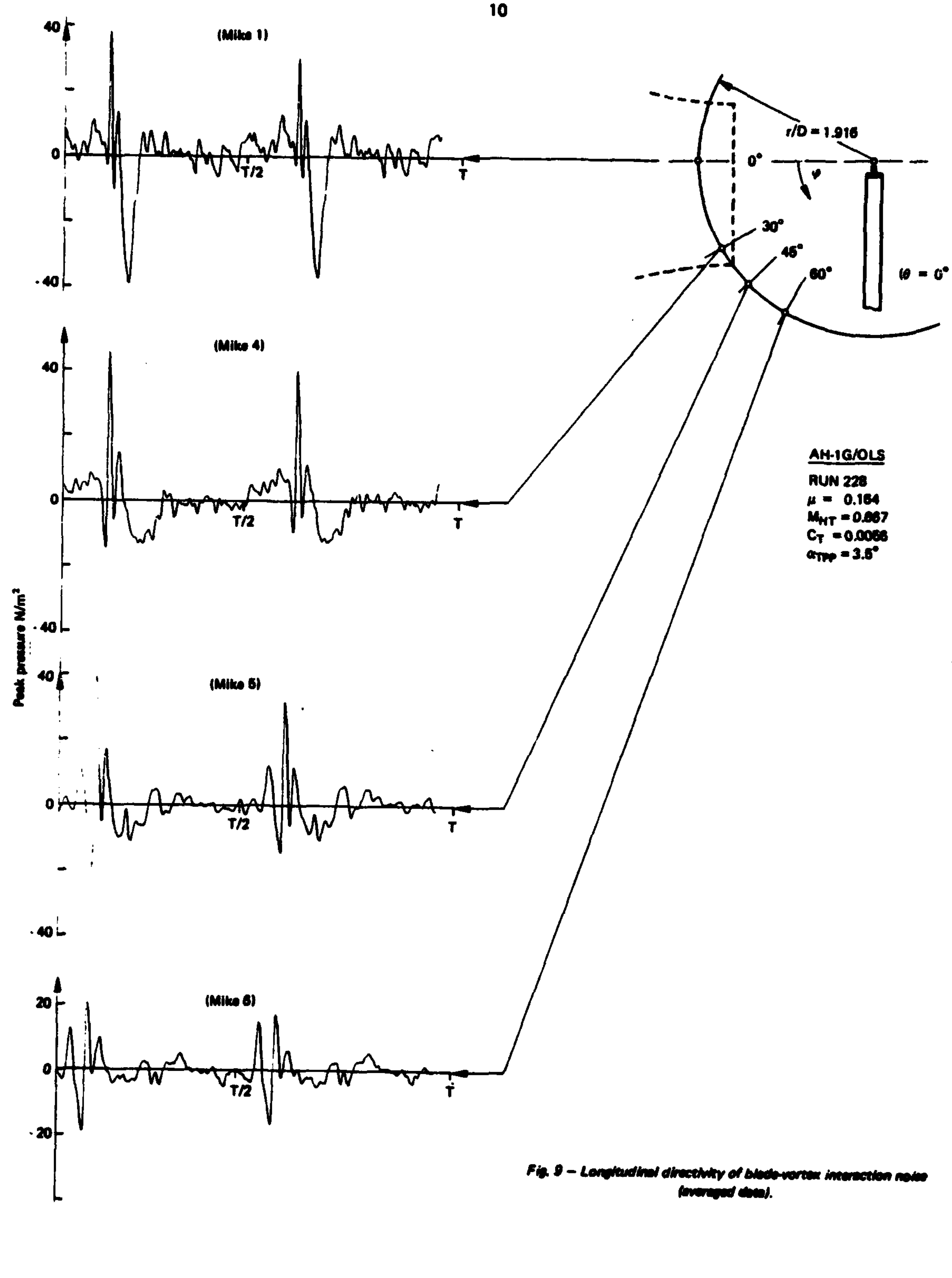

Fig. 9 – Longitudinal directivity of blade-vortex interaction noise (averaged data).

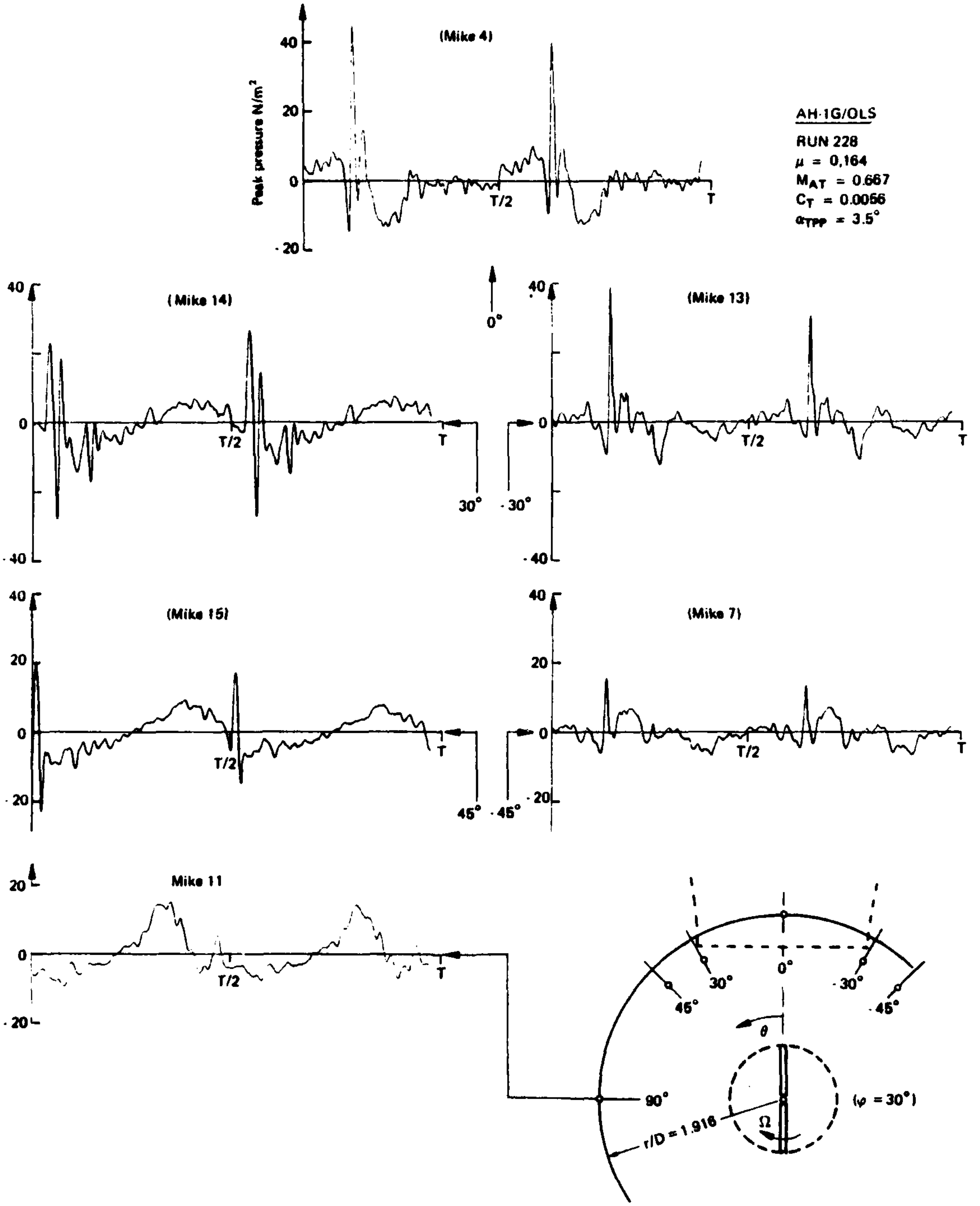

Fig. 10 – Lateral directivity of blade-vortex interaction noise (averaged data, $\varphi = 30°$).

Concluding Remarks

The data presented in this paper have shown that blade-vortex interaction noise on a single rotor two-bladed helicopter is a scalable event. Model rotor blade acoustic time histories are in general agreement with full scale data when acquired under similar non-dimensionalized conditions. Advance ratio, hover-tip Mach number, thrust coefficient, and local inflow ratio are the important non-dimensional scaling parameters in addition to local geometric scaling.

For the advance ratio of .164 which is representative of full scale flight at 60 Kts (IAS), the model-scale acoustic data are similar to the full scale acoustic data over a range of tip-path-plane angles representative of level flight to rates of sink of 600 ft/min. As inflow through the rotor disc changes, the respective levels of the blade-vortex interaction pulses change accordingly. In general, the more positive tip-path-plane angles (representative of descent) generate large acoustic pulses which occur earlier in time when measured from the $\Psi = 0°$ azimuthal position.

The peak amplitude of the advancing blade-vortex interaction pulses typical of a full scale 60 knot descent become a maximum about 30-to 40 degrees beneath the rotor tip-path-plane in the direction of forward flight. Pulse amplitudes are also a maximum from +30° to -30° to either side of the flight path.

Tunnel velocity variations (magnitude and direction) and the quality of the acoustic field near the important microphone measurement locations are two of the most important considerations of model rotor acoustic testing. Unsteady variations in w, u, and v cause the relative positions among the tip vortices and the rotor blades to change, making the blade-vortex interaction noise generation process unsteady.
The results of this paper suggest that careful aerodynamic design is needed to keep all components of the velocity variation below 1.0% of the free stream value. Special consideration should also be given to the acoustic design of the open jet nozzle. Because many of the important helicopter noise mechanisms direct noise forward along the flight path, microphones are often mounted in this direction. Unfortunately, these important measurement locations are generally not in an anechoic environment. Reflection and reverberation effects of the untreated nozzle degrade the acoustic qualities of the room and can make quantitative interpretations difficult.

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